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JPL 25-FOOT SPACE SIMULATOR SOLAR PERFORMANCE AND MARINER TEST RESULTS COMPARED TO FLIGHT DATA

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I. THE SOLAR SIMULATION SYSTEM CHARACTERISTICS COMPARED WITH THE SOLAR ENVIRONMENT IN SPACE

The JPL 25-ft space simulator solar simulation system has recently been modified[1] to concentrate all of the available radiant energy in a light beam of sufficient size and intensity to test the Ranger and Mariner spacecraft The properties of the beam compared to those of solar radiation in space are as follows:

	Simulator	True solar
Intensity	130 w/ft^2 (one Earth constant) to 170 w/ft^2	130-170 w/ft^2
Uniformity	±10% across the flats of a 5-ft (ID) hexagon	±0%
Spectrum	Hg-Xe arc	solar
Collimation	5.2 deg (half angle)	1/4 deg (half angle)

Several spacecraft system tests have been completed using the solar system in combination with vacuum levels in the 10^{-6} mm Hg range and tank wall shrouds cooled to liquid nitrogen temperature of -320°F. Some of the results obtained will be discussed later in the report.

II. DESCRIPTION OF THE SOLAR SIMULATION SYSTEM

Figure 1 is a schematic drawing of the JPL 25-ft space simulator optical system. The shaded portion indicates the light beam path from

[1]Mr. Norman Riise of JPL was instrumental in the development of the modification concept and supervised its execution.

the source lamp array through the optic system to the test specimen. Figures 2 through 7 are photographs of the actual optical elements in the system.

The "headlight" assembly (Fig. 2), which is the first element in the beam path, consists of a 2 1/2-kw HgXe compact arc lamp with a 16-in.-diameter parabolic glass mirror mounted above the lamp. A hemispherical Pyrex mirror is mounted below to collect light energy emitted from the bottom half of the lamp. The alignment of the entire unit is referenced to a mounting ring shoulder at the top of the 16-in parabola. This arrangement will enable an easy "headlamp" unit replacement when a new lamp is required.

Figure 3 shows several of the 19 stainless-steel plane mirrors, each 32 in. in diameter, used to form the external pseudoparabolic section. Each plane mirror receives light from seven headlamps. The pseudohyperbolic assembly (Fig. 4) consists of 19 slightly concave stainless-steel mirrors, each about 7 in. across.

The quartz window lens (Fig. 5) is convex on top, flat on the bottom (the vacuum side), and is said to be the largest quartz lens ever built.

The virtual source unit (Fig. 6), with its 290 individual parabolic concave stainless-steel mirrors, is the most unusual of the optical elements. The entire assembly is about 30 in. in diameter and is water-cooled throughout. Each small mirror reflects light to the entire portion of the 25-ft inside parabola which is in use. This feature is utilized to convert a nonuniform incoming beam from the window lens to a reasonably uniform beam upon arrival at the lower parabolic mirror. The virtual source is tilted off the axis of the chamber so that the light beam reflecting downward from the lower parabolic mirror clears the virtual source. The original on-axis design involved severe problems in attempting to compensate for the virtual source shadow. The virtual source assembly is supported on Invar tie rods in order to eliminate thermal distortion.

The 25-ft-diameter parabolic mirror on the inside of the chamber is made of 324 individual sections. Each section is spherical in optical contour and selected to approximate closely the overall parabolic shape desired. A portion of the parabolic mirror is shown in Fig. 7. Figure 7 also includes a photograph of the Mariner 3 spacecraft mounted in test position with a single train of seven lamps illuminating the test area. This spacecraft was a flight-ready spare identical to the successful Mariner 2 Venus probe spacecraft.

III. CALIBRATION OF THE SOLAR SIMULATION SYSTEM

The solar beam is hexagonal in cross section, measuring 5 ft across the hexagonal flats. The segment of the beam used for testing extends from 5 to 15 ft above the floor of the simulator. The beam was calibrated by measuring radiant energy distributions at various heights throughout the test volume. Figure 8 shows the traverse device which was used to scan the solar beam. The energy-sensing element was a silicon solar cell (3/4 by 3/8 in.) calibrated against an Eppley thermopile. Readings were made after the lamps had been set up to rated power (2500 ±100 w) and had stabilized for 2 hr.

Figure 9 shows the energy distributions measured at levels of 5, 10, and 15 ft above the floor of the simulator. The average intensity level in the test area was 170 w/ft^2, ±10%. This level is 30% in excess of the energy of the Sun at the orbital distance of the Earth (130 w/ft^2). The data were recorded under atmospheric conditions and were later verified under vacuum and cold-wall conditions.

It is interesting to note that the intrinsic quality of the beam in terms of uniformity is ±5% over the central 4-ft area. If a new virtual source were shaped to illuminate an area roughly 1 ft larger than the present beam size, the average intensity would be reduced to roughly 130 w/ft^2. The uniformity over the present 5-ft test area, however, would be improved from ±10 to ±5%. This example shows that a trade-off of area of illumination vs. intensity and/or uniformity can be made. It is also possible to trade intensity for collimation by optically "stopping down" the virtual source. This process can be accomplished by placing a circumferential mask on top of the virtual source, thereby reducing the effective source diameter.

The ability to mutually adjust the basic parameters of intensity, uniformity, and collimation is considered to be a very important system characteristic. This flexibility provides the opportunity to optimize the parameter relationships to best fit the requirements of a particular test program.

IV. COMPARISON OF FLIGHT TO GROUND-TEST DATA FOR THE *MARINER* VENUS PROBE

During September through December, 1962, consistent temperature distributions of the *Mariner 2* Venus probe in flight were obtained by telemetry. In January, 1963, the *Mariner 3* was tested in the 25-ft space simulator under simulated flight conditions. The *Mariner 3* spacecraft tested was a flight-ready spare of identical design to the *Mariner 2* used in the successful Venus mission

Since the 5-ft solar beam was not large enough to illuminate the extended solar panels, it was necessary to provide dummy thermal panels (see Fig. 8). The temperature of the dummy panels was controlled during the test to match the temperatures measured in the flight to Venus.

The space simulator test conditions were as follows:

Solar intensity	130 w/ft^2-(one Earth constant, or first day of mission) to 169 w/ft^2 (66th day of mission)
Vacuum	10^{-5} mm Hg or lower
Cold wall	-300°F

Table 1 summarizes the modes of operation of the spacecraft and the corresponding test conditions.

Table 2 presents the test data compared with the corresponding flight temperature data. During the test phase corresponding to flight near the Earth, the average bus temperature was 10°F low. Temperatures at other points varied from 3°F low to 13°F high. For a later period in flight, when the intensity was 169 w/ft^2 (66 days out), the average bus temperature was 22°F low, while temperature differences at other points varied from 13 to 28°F low. The reason for the larger disagreement at the higher solar intensity has not yet been determined. One possibility under study is that the reflective characteristics of the spacecraft surfaces may have deteriorated under the 66-day space exposure such that more solar energy was absorbed in flight than in the simulator. Although it would have been reassuring to have achieved better agreement with flight results, the test measurements are considered very useful as empirical design data.

These encouraging test results, coupled with recognition that the spacecraft temperature-control problem is very difficult to treat analytically, have contributed to the establishment of a proof-test policy. Future JPL spacecraft designs will be proof-tested in a space simulator having a performance capability equal to or better than the JPL 25-ft space simulator.

V. PLANS FOR IMPROVING THE SOLAR SIMULATION SYSTEM PERFORMANCE

At the present time there is a need to improve the level of performance described in Sections I and III of this report in two general areas:

1. The present performance is acceptable for Ranger and Mariner (Agena-class) spacecraft. However, a factor of 2 improvement in the collimation parameter would be very desirable (5.3 to 2 1/2 deg).
2. The Surveyor Project requires an area of illumination approximately 8 ft in diameter, with a solar intensity of 1 Earth constant (130 w/ft^2). (Since it is probable that no other U.S. space simulator now being planned or built will achieve this capability by December 1963, the Surveyor Project Office wants the option to proof-test the Surveyor spacecraft at JPL.)

In view of these requirements, the Jet Propulsion Laboratory has been investigating various means of improving the overall efficiency of the solar system. Two general plans showing promise are:

Plan A

Increasing the reflectivity of the mirrors in the system (Figs. 3, 4, 6, and 7).

Plan B

Increasing the power delivered to the system at the lamp source (Fig. 2).

Plan A, to improve the reflectivity, would replace the present metal surface mirrors with glass surfaces. An experimental program is underway at JPL to prove the feasibility of producing glass-surfaced mirrors which can be cooled adequately to withstand the high radiant-energy flux input without overheating and degrading the aluminized mirror surface. Test results to date are very encouraging. A comparison of the present metal mirror reflectivities with that which should be attainable with glass-surfaced mirrors (0.9) promises a potential overall efficiency improvement of 250%.

The Plan B methods for increasing the power delivered to the system at the lamp source are to improve the efficiency of the light-collecting reflector and to substitute different lamps which can supply more energy to the system.

The present maximum energy delivered is 130 w/lamp measured above the virtual source. This is only 10.5% of the 1250 w/lamp which is theoretically available at the lamp as useful radiant energy. Therefore, it appears technically reasonable to expect an improvement by optimizing the design of the collecting reflector.

In order to evaluate the improvement potential due to substitution of lamps, several different types of lamps have been tested in the 25-ft optical system. The power increase measured is shown in Table 3.

An independent study, which also involves the substitution (or mixing) of lamp sources, has been conducted for the purpose of optimizing the spectrum of the solar beam to match the Johnson curve. A summary of the results of this study is as shown in Table 4.

These study results show that a ratio of 1/3 HgXe to 2/3 Xe lamps is even superior to the carbon arc source which is generally considered to have the best single source spectral match to the Johnson curve.

This result, coupled with the fact that the Xe lamps are more efficient than the present HgXe lamps, indicates that a mixture will increase the energy delivered to the system and at the same time improve the spectrum. Specifically, if 87 lamps of the present 2.5-kw HgXe were replaced with 2.5-kw Xe lamps, the total energy increase should be 29%.

In summary, the improvements described in Plans A and B could increase the overall system efficiency by a factor of 2 to 3. It is necessary to increase the present efficiency by only 10% to satisfy the Surveyor requirement for an 8-ft test area at 130 w/ft^2. If the full potential efficiency improvement of 3 is realized, the present 5-ft beam collimation should improve from 5.3 to roughly 3 deg, which would be desirable for Ranger and Mariner. (It was noted in Section III that it is possible to trade off intensity for collimation by optically "stopping down" the virtual source.)

In conclusion, it appears that there is a high probability of achieving the level of performance currently required for the JPL flight programs.

VI. CONCLUDING REMARKS

1. The JPL 25-ft space simulator, equipped with the 5-ft solar simulation system, is considered to be the best experimental design tool available to proof-test the Ranger and Mariner spacecraft thermal control systems. A Mariner 3 spacecraft was thermally tested in this facility, yielding data which agreed acceptably with flight data obtained from Mariner 2 enroute to Venus. The Jet Propulsion Laboratory has established a requirement that future spacecraft systems are to be tested in a simulated space environment equal to or better than the JPL 25-ft space simulator.
2. There is a high technical probability of achieving the improved solar simulation performance level required for testing spacecraft of the Surveyor and Mariner B (Centaur) class in time to meet present flight program test schedules. This also means that the collimation angle of the present 5-ft Ranger/Mariner solar beam can be substantially decreased.
3. The Bausch and Lomb solar simulation system, as modified by JPL, became operational in January 1963. It has a unique capability and as such represents a "first" in Sun simulation technology for NASA and

JPL. The 5-ft system is the largest operational, well-collimated system known to exist at the present time.

4. It is a well-known fact that solar simulation technology is in its infancy. Much needs to be learned in order to define an acceptable quality of simulation without becoming unnecessarily extravagant. In view of these uncertainties, the performance flexibility of the JPL system is considered to be very valuable. The important characteristics of flexibility are:

    1. The ability to trade off three major parameters of solar simulation consisting of intensity, area of illumination, and collimation. This flexibility permits the optimization of Sun simulation to best fit the particular testing requirements of a given spacecraft or spacecraft component.

    2. The ability to mix lamp sources to tailor the spectral distribution of energy to fit the Johnson curve. An alternate approach is to provide a spectral distribution which will provide the same thermal input to a specific spacecraft configuration as the Sun. Stated differently, the goal can be to simulate the <u>effect</u> of the Sun as opposed to <u>duplicating</u> solar radiation spectrum.

Table 1. Mariner flight conditions simulated in ground tests

Mode	Days from launch	Spacecraft condition	Solar intensity, w/ft^2
I	1	Cruise, science off	127
II	13	Cruise, science on	128
III	64	Cruise, science on	167
IV	66	Cruise, science off	169

Table 2. Comparison of Mariner flight temperature data to ground test data[a][b]
(°F except as shown)

	Mode I		Mode II		Mode III		Mode IV	
	MR-3 test	MR-2 flight	MR-3 test	MR-2 flight	MR-3 test	MR-2 flight	MR-3 test	MR-2 flight
Hex temperature (average)	71	81	75	85	85	105	80	102
Case I science	70	76	79	88	92	112	81	106
Case II communications	79	86	82	92	93	115	89	113
Case III data encoder and command electronics	80	86	82	92	93	112	90	111
Case IV attitude control/CC and S	68	76	69	76	77	91	74	90
Case V power	70	88	72	86	80	105	75	103
Case VI battery	62	72	66	78	77	97	71	92
Booster regulator	69	82	74	90	83	102	72	94
Prop tank	70	76	73	80	85	101	81	93
Earth sensor	80	88	80	90	58	83	55	83
A/C nitrogen	73	70	74	76	90	109	84	108
Upper thermal shield	69	56	69	62	84	93	80	93
Internal hex power (w)	139	137	148	152	146	149	130	130
Solar intensity (w/sq ft)	127	127	128	128	167	167	169	169

[a]The temperature data shown above were provided by D. W. Lewis, who was the Test Director in charge of the Mariner test. Mr. Lewis is Group Supervisor of the Thermal Control Group in the JPL Spacecraft Development Section (Section 352).

[b]MR = Mariner

Table 3. Lamp evaluation summary

Lamp	Measured power above lens, W	Estimated power with cup, +54%	Improvement ratio over present 2.5 HgXe lamps
2.5-kw HgXe	163 (with cup)		1.0
2.2-kw Xe	206 (with cup)		1.27
2.5-kw Xe	not available	234	1.44
5.0-kw HgXe	151 (without cup)	232	1.42
5.0-kw Xe	262 (without cup)	400	2.46

Table 4. Absorptivities of several materials to different arc lamp spectra

Material	Absorptivity	$\frac{\text{Absorp. to lamp spect.}}{\text{Absorp. to solar spect.}}$			
		Carbon arc	HgXe	Xe	1/3 HgXe 2/3 Xe
Polished aluminum	0.235	0.97	0.99	1.00	1.00
Aluminum mirror	0.100	0.97	0.93	1.05	1.02
Gold	0.226	0.86	1.32	0.76	0.94
Aluminum silicon resin paint	0.247	0.98	0.98	1.02	0.99
Zinc oxide silicon paint	0.177	0.96	1.39	0.84	1.01

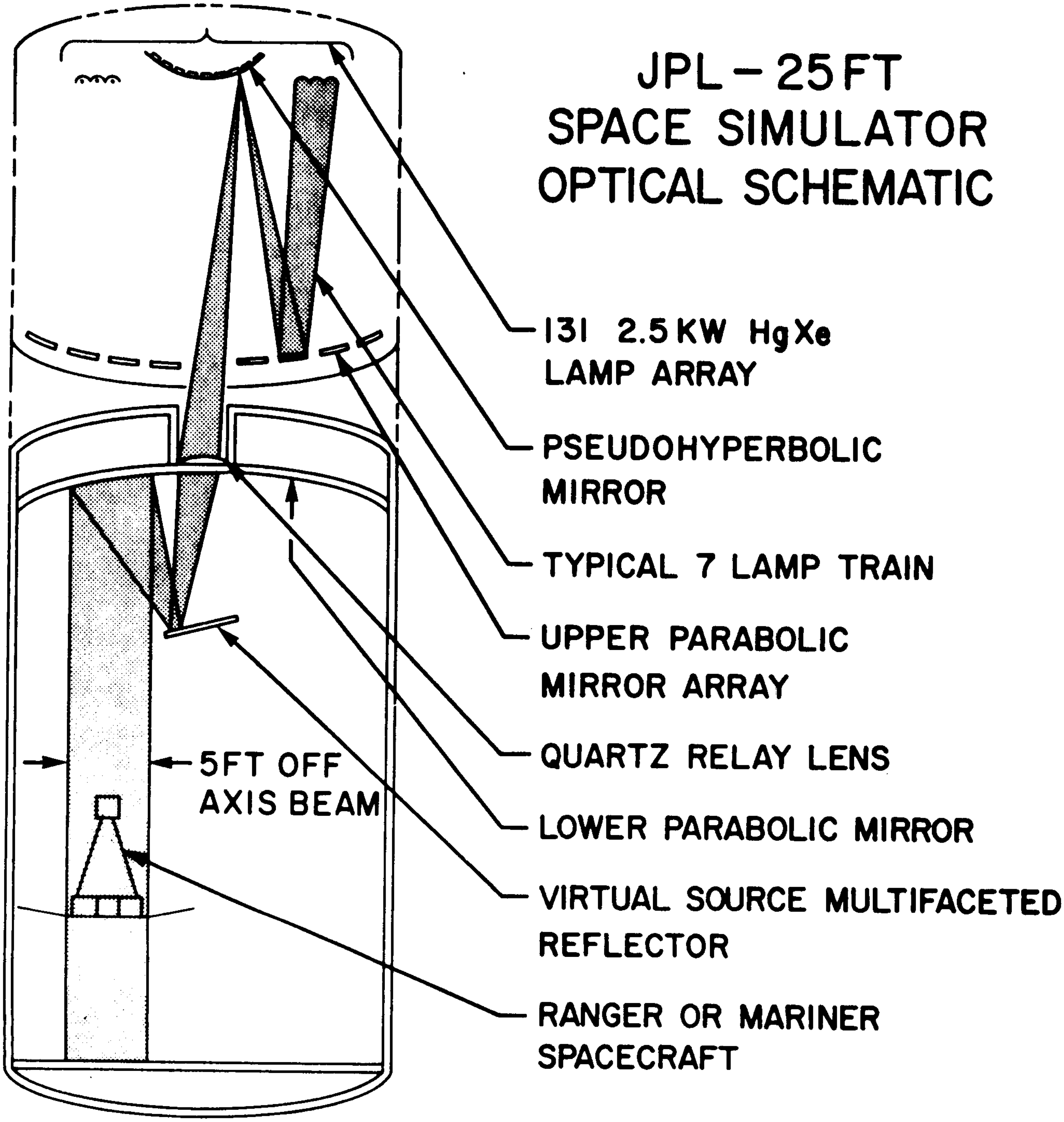


Fig. 1. JPL 25-ft space simulator optical schematic

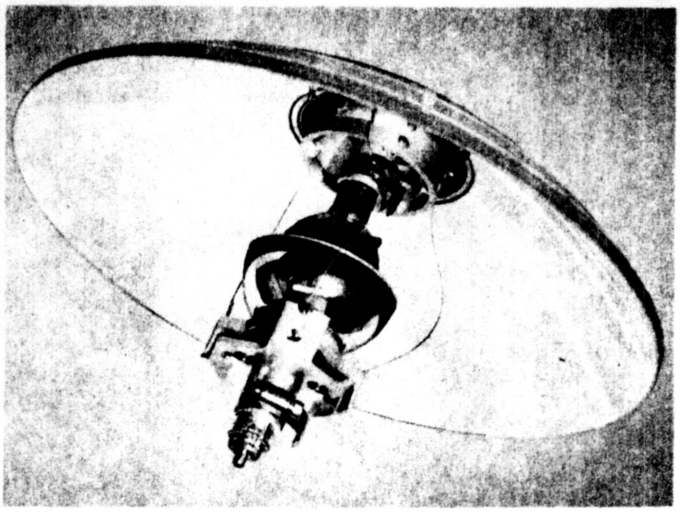


Fig. 2. Headlight assembly

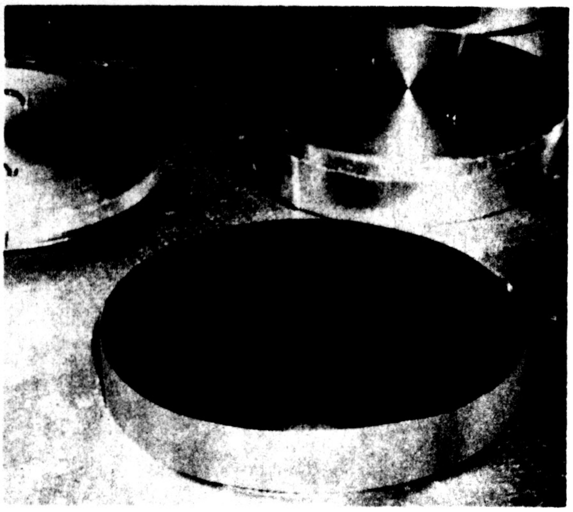


Fig. 3. Stainless-steel plane mirrors

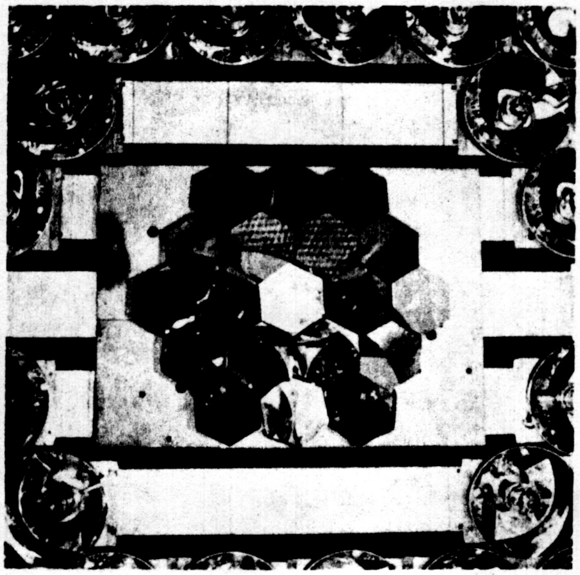


Fig. 4. Pseudohyperbola

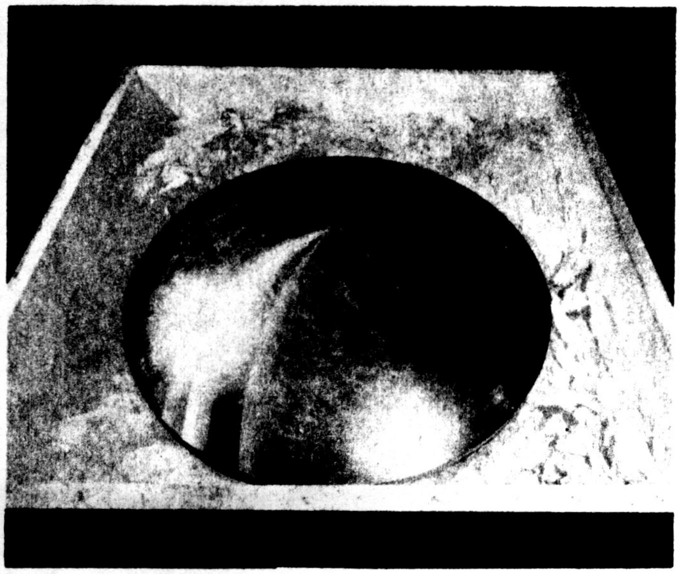


Fig. 5. Quartz window lens--36 in. diameter (in shipping crate)

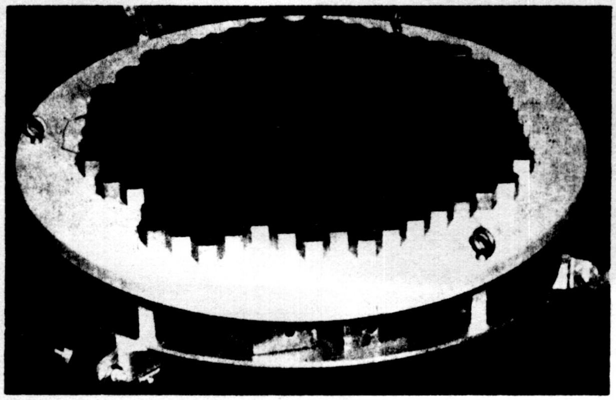
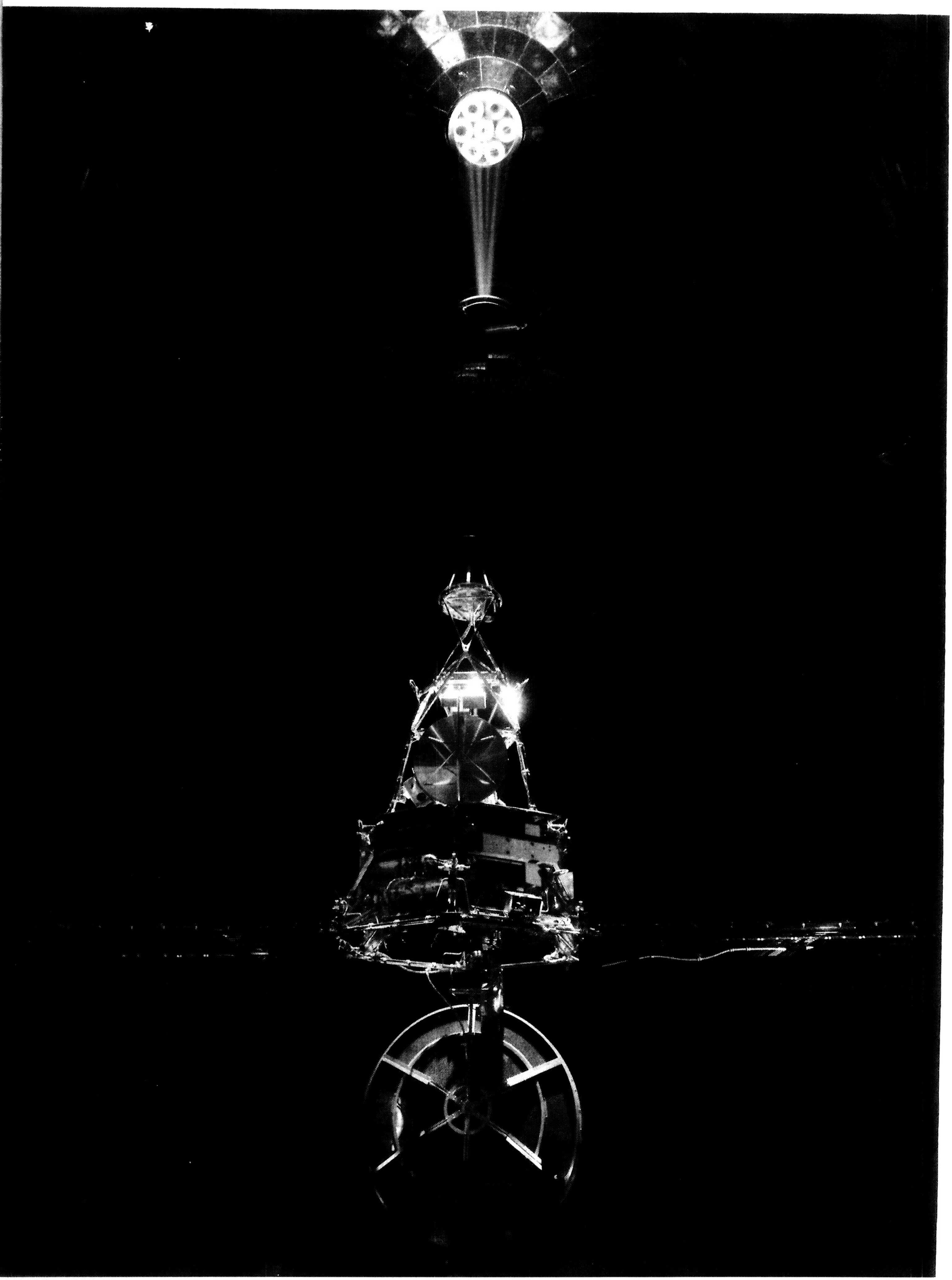


Fig. 6. Virtual source



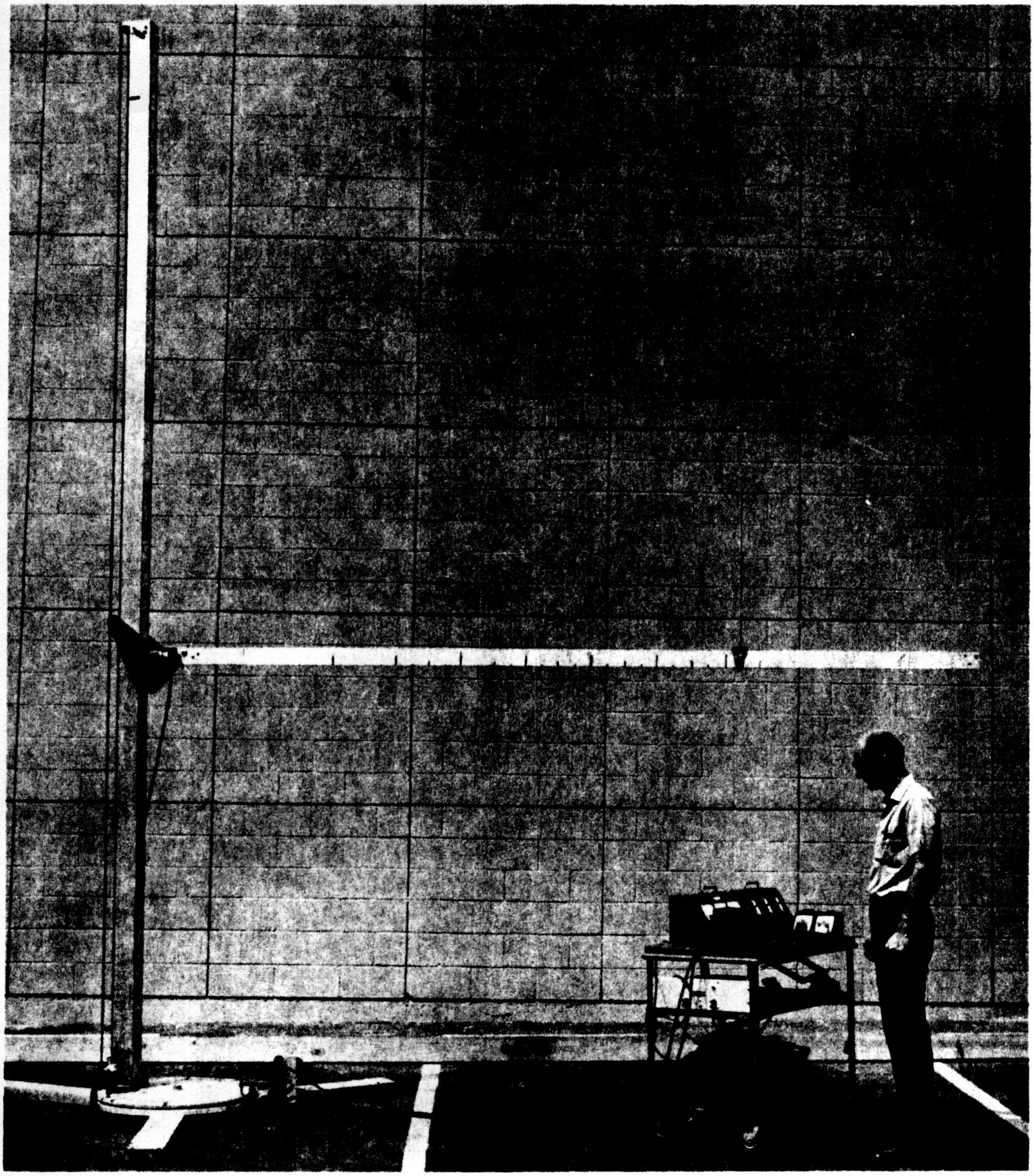


Fig. 8. Intensity measurement traverse rig

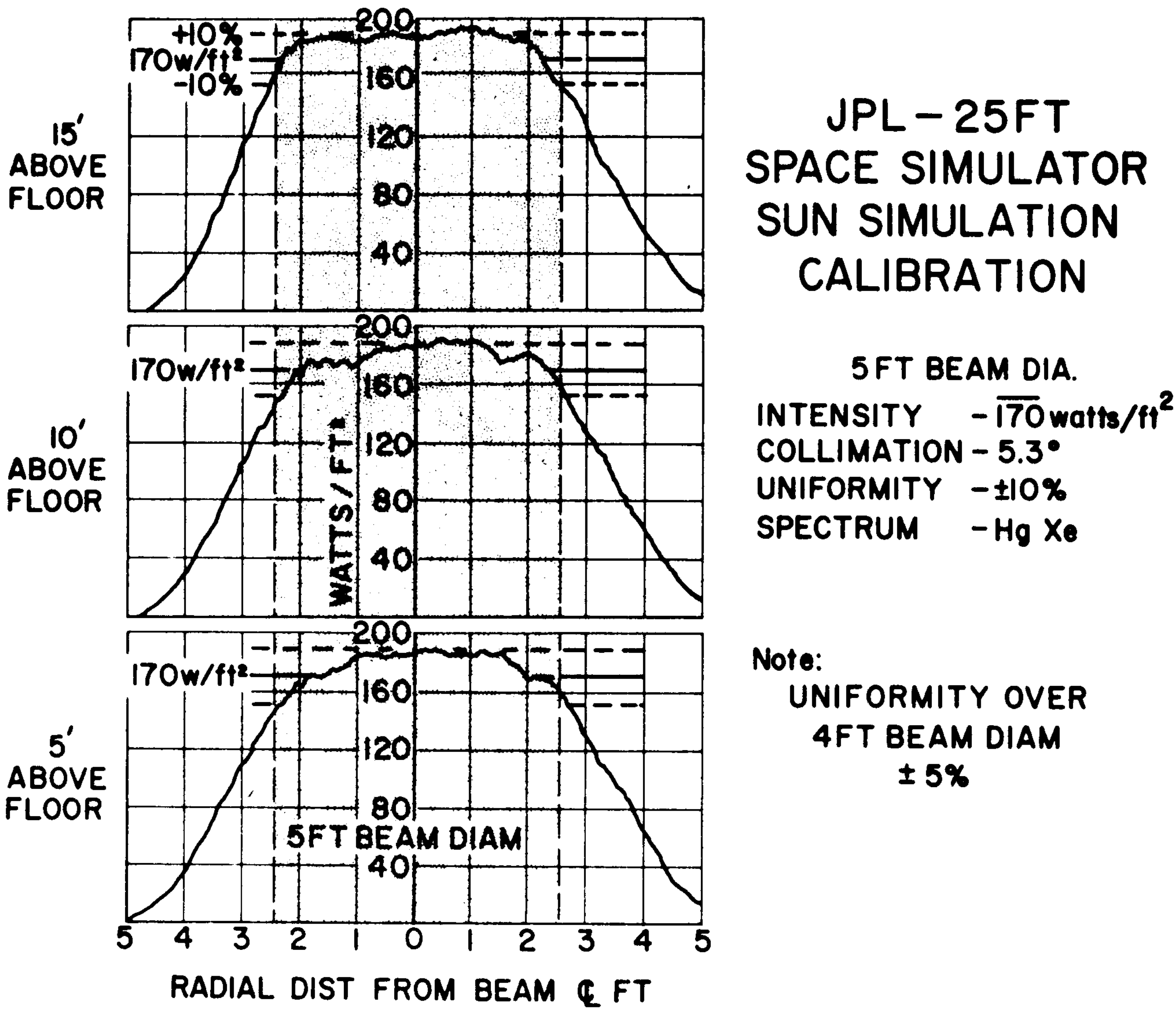


Fig. 9. JPL 25-ft space simulator Sun simulation calibration